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Paper No. 42

ABLATIVE HEAT SHIELD DESIGN FOR SPACE SHUTTLE

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Information in this paper resulted from work conducted under NASA Contract NAS1-11592.

ABSTRACT

Ablator heat shield configuration optimization studies were conducted for the Orbiter. Ablator and RSI trajectories for design studies were shaped to take advantage of the low conductance of ceramic RSI and high temperature capability of ablators. Comparative weights were established for the RSI system and for direct bond and mechanically attached ablator systems. Ablator system costs were determined for fabrication, installation and refurbishment. Cost penalties were assigned for payload weight penalties, if any. The direct bond ablator is lowest in weight and cost. A mechanically attached ablator using a magnesium subpanel is highly competitive for both weight and cost.

INTRODUCTION

Ablators are a well-established system of thermal protection having been employed on vehicles such as Apollo, Gemini, Viking, X-15, Titan, PRIME and others. The need for ablator refurbishment following each flight is a drawback of this thermal protection system and has led NASA and industry into the development of reusable ceramics (RSI). The RSI system of thermal protection has been baselined for use on the Shuttle Orbiter.

Much work needs to be done to qualify the RSI for space shuttle application and, to quote E. S. Love from the Tenth Von Karman Lecture, "Ablators offer a confident fall-back solution (temporary) for both leading edges and large surface areas, should development of the baseline approaches lag."

In the past, ablator systems have been bonded directly onto the structures they are designed to protect. While this approach is both low in weight and cost effective, it has the drawback for the Shuttle Orbiter of taking up critical turn-around time for refurbishment between flights, and during refurbishment creates a problem of debris and dust control.

The purpose of this program was to investigate Shuttle Orbiter TPS design concepts utilizing available state-of-the-art ablators. An end objective of the program was to obtain ablator TPS weight and cost estimates based on detailed, verified heat shield designs. RSI cost estimates were not part of this program. Direct bond ablator and RSI designs were prepared for weight comparison purposes. The bulk of the effort has dealt with methods of mechanically attaching prepared ablator panels onto the Orbiter. Radiant heat tests were conducted to verify the design concepts.

The program was divided into five tasks:

Task 1 - Design Criteria;

Task 2 - Flight Environment;

Task 3 - Heat Shield Designs;

Task 4 - Design Verification;

Task 5 - Weight and Cost Analysis.

DESIGN CRITERIA (TASK 1)

Design criteria was prepared in order to develop valid heat shield configurations for the total environment from prelaunch through reentry and landing. Criteria was prepared for the trajectory definition and for the thermal and stress analysis.

Factors of Safety

Safety factors on loads and pressures for prelaunch through deorbit are based on engineering practice developed for boosters and spacecraft and for entry and atmospheric flight, and are those commonly used in the design of aircraft.

FACTORS OF SAFETY			
Location	Ultimate Design Condition	Ultimate or Design Factor of Safety (X)	Limit Load or Nominal Heating Load
Total Orbiter	Through Orbit	1.4	Limit Load
Total Orbiter	Entry	1.5	Limit Load
Leading Edges and Lower Forward Surfaces	Ascent and Entry	1.15	Nominal Heating Load
Lower Surfaces Aft		1.25	
Upper Surfaces		1.50	

Backface (Structural) Temperature Limits

Ablator Bond Line	500°F
Maximum Structural Temperatures	
At Start of Entry	100°F
At Completion of Entry	350°F

Ablator Strain Limits

Ablators are low strength (< 70 psi at room temperature) and low modulus (< 3,000 psi at room temperature) materials in the virgin form with unspecific mechanical properties after charring. The following induced strain limits have been developed for the Viking program:

	Virgin	Charred
Tension Strain (in./in.)	1.0%	0.6%
Compression Strain (in./in.)	1.0%	1.0%

Strength Analysis

The ablation material shall not be considered load carrying but shall be included in thermal and mechanical deflection analysis to determine the strain in the ablation material. The subpanel shall be capable of carrying design loads without the ablator, and without exceeding the following surface waviness deflection criteria:

H = 0.0125L Limit

L = Panel Wave Length;
H = Maximum Deflection (wave height)

FLIGHT ENVIRONMENT (TASK 2)

Orbiter flight environments were defined for the boost, orbit and entry conditions. Airloads and thermal histories were established including interference heating during ascent. Heating load distributions accounted for the variances in time at which transition from laminar to turbulent flow occurs. Flow transition near the fuselage nose and wing leading edge occurs several hundred seconds following peak heating. Transition from laminar flow was considered to occur when

$$\frac{Re_\theta}{M_L\left(\frac{Re_L}{X}\right)^{0.2}} = 10$$

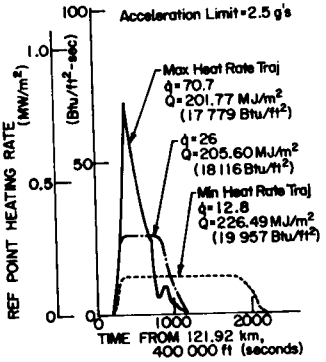
where: Re_θ = local Reynold's number based on momentum thickness
M_L = local Mach number
Re_L/X = local unit Reynold's number

Fully turbulent flow was assumed to occur at a location twice the length of the transition onset length.

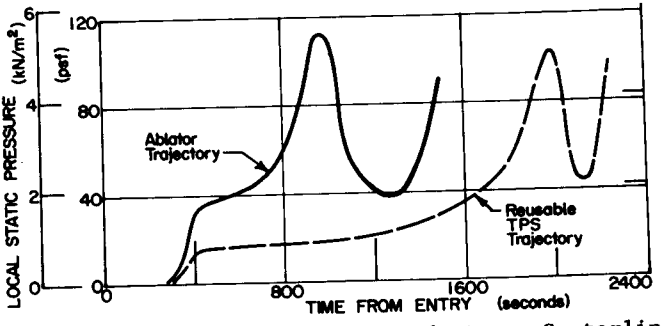
The Orbiter heating load distributions were normalized to reference point selected as 50 feet aft of the fuselage nose on bottom centerline.

Trajectory Shaping

Design entry trajectories were established for the ablator TPS and the RSI TPS. The ablator trajectory was shaped to take advantage of ablator's high heating rate capability. The maximum heat rate trajectory within the acceleration limit of 2.5 g's yields a $\dot{q}$ = 70.7 BTU/ft^2-sec at the reference point. With a thermal design factor of safety of 1.15, the design heat rate equals 81.3 BTU/ft^2-sec. Since SLA-561 ablator is limited to a maximum heating rate of 60 BTU/ft^2-sec, the entire lower surface of the Orbiter would require the higher densities of ESA-3560HF and ESA 5500. Reducing the entry angle and heating rate to 26 BTU/ft^2-sec at the reference point permits use of low density SLA-561 over 98% of the Orbiter surface area. The $\dot{q}$ = 26 BTU/ft^2-sec at the reference point was therefore designated as the ablator design trajectory. The RSI trajectory was shaped to take advantage of the ceramic TPS low thermal conductivity for total heat insulation. By reducing entry angle to the skip-out limit, heat rate is minimized and total heat maximized resulting in an efficient RSI TPS design. Local static pressure histories were prepared for the Orbiter for ascent and entry. These pressures are used for the ablator subpanel design.

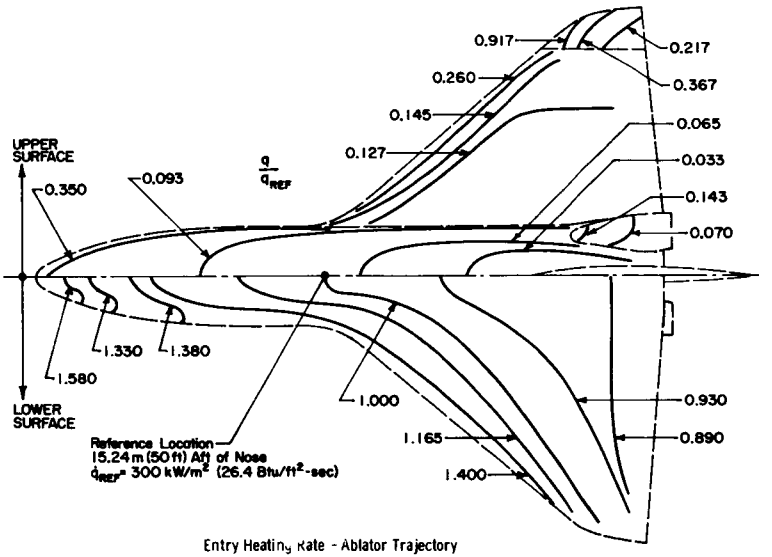


Trajectory Shaping

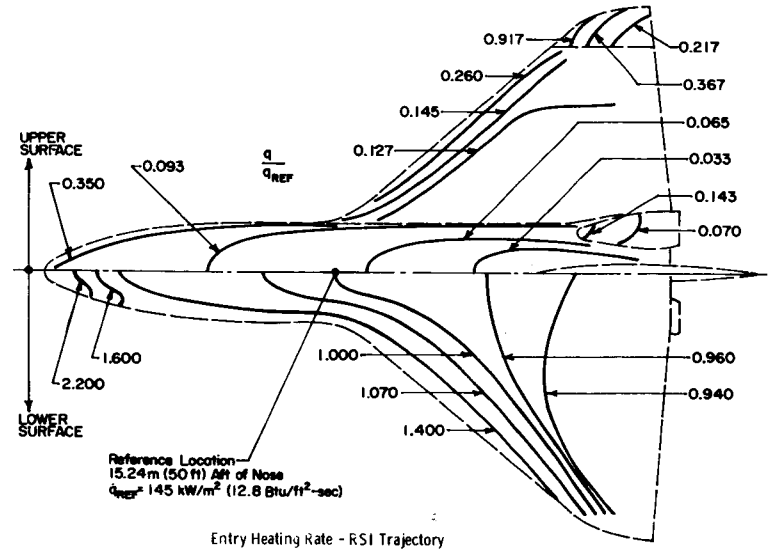


Entry Local Static Pressure History at Body Lower Centerline Reference Location

Heating Rate Distributions--Heating rate distributions normalized to the reference point at the lower fuselage centerline 50 feet aft of the fuselage nose were prepared for the ablator and RSI trajectories.



Entry Heating Rate - Ablator Trajectory



Entry Heating Rate - RSI Trajectory

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HEAT SHIELD DESIGNS (TASK 3)

The contractual program investigated eight locations on the Orbiter to establish heat shield configurations, weights and costs. This paper will concern itself with one location--the reference point, 50 feet aft of the fuselage nose on the lower surface centerline.

The baseline method of ablator attachment was considered to be the direct bond. Alternate configurations investigated were:

1. Ablator bonded to a subpanel plate and mechanically attached directly to the Orbiter structure;

2. Ablator bonded to a subpanel honeycomb panel and mechanically attached directly to the Orbiter structure;

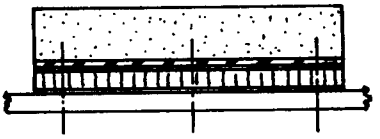
3. Ablator bonded to a subpanel honeycomb panel and mechanically attached through standoff fittings to the Orbiter structure.

Plate Subpanel

Aluminum 2024-T81
Lockalloy
Magnesium HM-21A

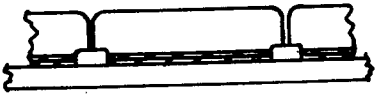
Ablator Direct Bond

Baseline



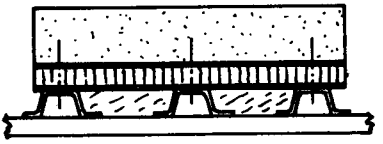
Honeycomb Subpanel

Aluminum 2024-T81
Magnesium HM-21A
Graphite Polyimide



RSI Direct Bond

Through Strain Isolator



Honeycomb Standoff Subpanel

Aluminum 2024-T81
Graphite Polyimide

Each configuration was examined for 5, 10, 15 and 20 inch attachment spacing for weight and cost optimization. In addition, four materials of construction for the subpanel were studied: aluminum 2024-T81, Lockalloy, magnesium HM-21A, and graphite polyimide.

Thermal Analysis

Thermal analysis for ablator sizing was carried out with the Martin Thermochemical Ablation Program (TCAP III). Data input to this program includes trajectory data; i.e., velocity, altitude, heating rate, recovery enthalpy; thermophysical properties for the ablator material and backup structure materials; ablation kinetics; and geometry of the model being analyzed. Analysis results include time-temperature distributions throughout the model and a time-density profile through the ablative material.

Analysis for sizing the RSI material and for developing back-up structure modeling techniques was carried out with the Martin Three Dimensional Heat Transfer program. Data input and analysis results are similar to TCAP III with the exception that no considerations are made for an ablation process. Both programs allow for variations of conductivity with pressure as well as temperature.

Ablator thickness design charts were prepared for the ablator trajectory assuming an entry start temperature of $100^{\circ}F$ and limiting the maximum structural temperature after entry to $350^{\circ}F$. Given $\dot{q}/\dot{q}_{ref} = Q/Q_{ref}$, using the heating rate distribution charts and computing $\sum_{i}^{n} t \rho C_p$, the ablator thickness can be determined for any point on the Orbiter surface. $\sum_{i}^{n} t \rho C_p$ is the total heat capacity of the local Orbiter structure, the subpanel and the ablator bond line.

EXAMPLE: At the reference point, determine ablator thickness for an aluminum subpanel, t = .031

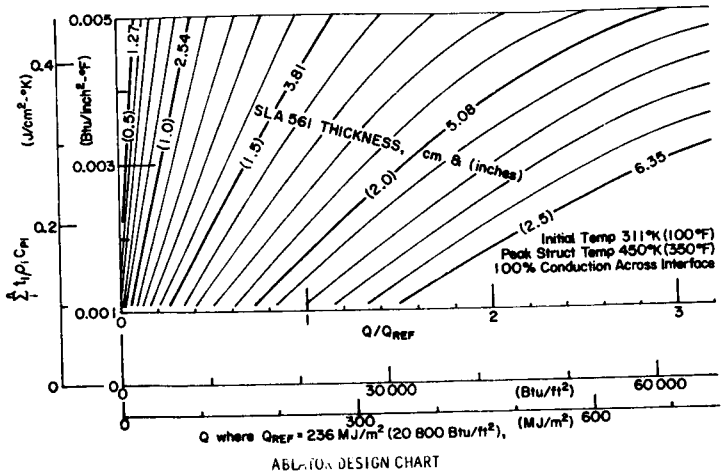
Orbiter Structure, .103 aluminum	$\rho = 0.100$; $C_p = .225$
Ablator Bond, .030 RTV	$\rho = 0.054$; $C_p = .300$
Subpanel, .031 aluminum	$\rho = 0.100$; $C_p = .225$

$$\sum_{i}^{n} t \rho C_p = (.103)(.100)(.225) + (.030)(.054)(.300) + (.031)(.100)(.255) = .00351$$

Enter the design chart with $\sum_{i}^{n} t \rho C_p = .00351$ and $Q/Q_{ref} = 1$ and read Thickness SLA-561 = 1.63 in.

Stress Analysis

The plate and honeycomb subpanels are designed by local aerodynamic airloads. Orbiter structural strains are isolated from



ABLATOR DESIGN CHART

the subpanels by using a mechanical fastener in an oversized hole. The strains and deflections of the subpanels must be limited to prevent induced strains in the ablator from exceeding 1% and to limit surface waviness to less than 0.0125 times the attachment spacing. Subpanel overall dimensions are limited by handling capabilities. A 42 inch by 42 inch panel size was selected for this program. Since total heat shield weight and cost optimizes with small attachment spacing (less than 10 inches), the analytical methods for internal loads predictions were selected from "Theory of Plates and Shells" by Timoshenko, and Woinowski-Krieger, "Bending of Plates Supported by Rows of Equidistant Columns."

Allowable Loadings--The subpanel plates are checked for thickness limitations to meet deflection requirements and moment limitations to meet ablator strain limits. Both of these are a function of support spacing.

$$\text{Deflection Limit} \leq 0.0125\ell$$

$$\text{From Timoshenko deflection} = \frac{.00581\ q\ \ell^4}{D}$$

$$\text{Combining and simplifying, } t \geq .0062\ell$$

$$\text{Ablator Strain Limit} \leq 1\% \text{ (during ascent)}$$

$$\epsilon = \frac{\text{Moment } (1 - \mu^2)(\bar{y})}{EI}$$

where $\bar{y}$ is distribution from neutral axis of plate to ablator surface and I is for plate only.

$$\text{Allowable moment} = \frac{(0.01)EI}{(1 - \mu^2)(\bar{y})} \quad \text{with } \epsilon = 0.01$$

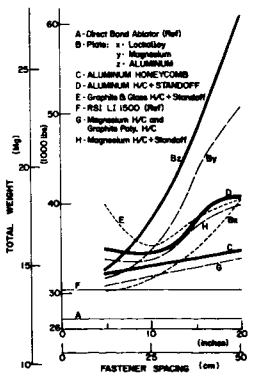
Honeycomb subpanels must also be checked for face wrinkling and intracell buckling.

$$\text{Face wrinkling moment} = (0.33)\left[(E_c)(E_F)(t_{core})\right]^{1/2}(t_{face})^{3/2}$$

Example Heat Shield Configurations at Reference Point

ℓ	Direct Attached Aluminum Plate 2024-T81			Direct Attached Aluminum Honeycomb 2024-T81				Standoff Aluminum Honeycomb 2024-T81			
	Subpnl. Thick-ness	Abl. Thick.	Crit. For	Face Thick.	Core Height	Abl. Thick.	Crit. For	Face Thick.	Core Height	Abl. Thick.	Crit. For
5	.031	1.74	∂	.005	.030	1.86	MIB	.005	.180	1.50	MIB
10	.062	1.64	∂	.005	.190	1.86	MIB	.007	.480	1.47	MAS
15	.148	1.51	MAS	.006	.300	1.85	MIB	.009	.900	1.44	MFW
20	.228	1.41	MAS	.007	.420	1.82	MAS	.014	.990	1.41	MAS

∂ Deflection
MAS Ablator Strain Moment
MIB Intracell Buckling Moment
MFW Face Wrinkling Moment



TPS Attachment Configuration Weight vs Fastener Spacing

Total Heat Shield Configuration Weights

Using the method illustrated for the reference point, heat shield configurations were determined for other materials and locations on the Orbiter and the total heat shield weights plotted against fastener support spacing.

DESIGN VERIFICATION (TASK 4)

A test program was conducted to evaluate the mechanically attached heat shield concept. The aluminum subpanel system at 10 inch fastener spacing was selected as being representative of most of the direct mechanically attached configurations. Testing was performed in three parts.

VERIFICATION TEST PROGRAM		
Test	Specimen	Objective
Ablator Panel Open Gap	SLA-561 Ablator on Aluminum Subpanel, 10 inch Fastener Spacing. Size: 22 in. x 44 in.; Gap Size: 0.070 in.	Verify: 1. Open Gap Concept 2. Fastener Design 3. Temperature Distribution in Fastener Cavity
Gap Sealer	Fiberfrax Rope, 1/4 inch diameter; Fiberfrax Blanket	Determine resiliency in thermal environment.
Ablator Panel Sealed Gap	SLA-561 Ablator on Aluminum Subpanel, 10 inch Fastener Spacing. Size: 22 in. x 44 in.; 0.80 inch Fiberfrax Blanket Sealer	Verify: 1. Sealed Gap Concept 2. Fastener Design 3. Temperature Distribution in Fastener Cavity

Open Gap Test--Ablator panels can be installed with a limited width gap at the joint between adjacent panels (1/8 inch for 42 inch x 42 inch panels). Due to the thermal coefficient of expansion of the ablator, the gap closes with increasing temperature limiting aerodynamic heating at the bottom of the gap. The entry heating pulse was simulated using radiant heat lamps. Although heating was not uniform, some gap closure with acceptable temperature limits at bottom of gap was noted. The attaching fasteners tended to bind, inhibiting free thermal motion of the subpanel. Temperature in the fastener cavity in the ablator was not excessive.

Gap Sealer Tests--Candidate materials were tested in ovens under deformation at elevated temperatures. Springback or recovery of deformation following temperature reduction was measured. Five percent to ten percent springback was desired following heating. Only the Fiberfrax blanket compressed normal to fiber direction was satisfactory.

Sealed Gap Test--Ablator panels can be installed with a sealer in the joint between adjacent panels. During ascent heating the gap closes partially, then opens while in orbit, closing again at entry. The selected sealer was Fiberfrax blanket, which is resilient enough to take these thermal cycles. The test panel was exposed to ascent and entry heating using radiant heat lamps. The sealer did not perform satisfactorily since temperatures were higher in the bottom of the gap than under the ablator. Fiberfrax blanket is stratified and the thermal conductivity is therefore transversely isotropic with the low values across the gap and the high values parallel to the gap.

WEIGHT AND COST ANALYSIS (TASK 5)

This program has developed design concepts and weight data for several competing methods for ablator attachment to the Orbiter structure. It remains only to select the best attachment system on a weight and cost basis. Competing heat shield weights were converted to costs and added to operational costs to establish the total program costs. The ablator configuration choice on the basis of minimum program costs thus includes the effect of heat shield weight.

Payload Weight Penalties

To aid in the optimum ablator heat shield selection, an ablator heat shield was assumed for the first five years or 151 flights of Shuttle operations--1979 through 1983. The payloads planned for that period were identified from the "Preliminary Traffic Model for the Space Shuttle", November 9, 1972 by MSFC. Each of the competing heat shield design weights was compared to the payloads of the traffic model and payload weight penalties, if any were determined. To ascribe a dollar value to these weight penalties, a cost per pound to orbit was derived. All program costs were apportioned against all the payload weight. A DDT&E cost of $5,150M was apportioned to 445 flights for a cost per flight of $11.57M or $1,747.5M for the first 151 flights. Added to this is an operational cost of $10.5M per flight or $1,585.5M for the first 151 flights. The total cost for the first five years of 151 flights is $3,333M. The first 151 flights carry a total of 4.56M lbs of payload. Dividing $3,333M by 4.56M lbs yields a cost per pound to orbit of $731/lb. This does not include the costs of the payloads themselves.

The payload weight penalties for each of the competing ablator systems was multiplied by this cost per pound to orbit of $731/lb and added to the operational costs of each ablator system to establish a total program cost. The RSI heat shield weight was used as a baseline to establish payload weight penalties. In many cases, the full payload weight capability of the Orbiter was not utilized due to volume constraints. This reduced payload was utilized in establishing payload weight penalties. For example:

THREE MISSIONS OF THE 1981 PERIOD			
Flight Number	15	18	20
Payload Capability	65,000	45,000	45,000
Actual Payload	-20,150	-17,100	-37,450
Unused Capability	44,850	27,900	7,650
RSI Heat Shield	30,240	30,240	30,240
Available Heat Shield Weight	75,090	58,140	37,890
Ablator Direct Attached by Aluminum Plate at 15"	-48,782	-48,782	-48,782
Payload Weight Penalty	0	0	-10,892
Payload Penalty Cost at $731/lb	0	0	$7.96M

All the payload penalties for all flights were added resulting in $364M cost penalties for the aluminum subpanel with 15 inch spacing.

Operational Costs--Operational costs are the fabrication, refurbishment and quality assurance tasks incurred during the 151 flight program. These costs include:

- Ablator slab raw materials and fabrication including scrappage (this accounts for 3/4 of the operational cost);
- Subpanel raw materials and assembly costs, including scrappage;
- Assembly of ablator slab to the subpanel;
- Installation of ablator panel assembly--tools and labor;
- Removal of used ablator panels--tools and labor;
- Repair materials and labor due to damage from handling during packing, shipping, storage and installation;
- Bond line inspection;
- Mechanical fastener inspection;
- Subpanel fabrication inspection;
- Refurbishment cleanliness inspection;
- Inspection for damage following ablator installation;

- Inspection of repaired areas.

The total operations costs show the direct bond heat shield and the aluminum and magnesium subpanel mechanically attached heat shields to be very competitive with the 10 inch spacing subpanels having the lowest cost primarily due to the reduction in ablative materials required and lower installation costs due to smaller numbers of fasteners.

Total Program Costs--The inclusion of the heat shield weight as a payload weight penalty costs has a direct effect on determining the optimum heat shield selection. The direct bond system has the lowest total program cost followed closely by the magnesium and aluminum subpanel plate systems at 5 inch fastener spacing. The low operational cost magnesium and aluminum subpanels at 10 inch spacing are prohibitively heavy as is shown by the $34.9M and $76.9M payload weight penalties. For the direct bond system, the effect of dust and debris and turn-around time was not evaluated.

Considering these items qualitatively, the slightly higher cost, direct attached magnesium and aluminum plate systems at 5 inch fastener spacing are attractive alternates. The selection of mechanically attached heat shields must be made prior to Orbiter CDR to permit inclusion of anchor nuts into the basic design. The addition of this fastener hardware following the original engineering release becomes increasingly difficult and at the point where Orbiter hardware has been fabricated without heat shield anchor nuts, adding them becomes prohibitive.

OPERATIONAL COSTS ($M) - FIVE YEARS - 151 FLIGHTS

Configuration		Ablator Slab	Sub-Panel	Panel Ass'y	Install. and Removal	Repair	Qual. Assur.	Total
Subpanel	Fastener Spacing							
Direct Bond	N. A.	125.7	-	-	27.9	4.4	6.8	164.8
Aluminum Plate	5 inches	120.7	1.2	8.7	21.9	6.6	9.2	168.3
Magnesium Plate		122.1	1.8	8.7	22.4	6.8	10.0	173.8
Lockalloy Plate		120.7	31.1	10.4	25.6	6.8	10.6	205.2
Aluminum Honeycomb		124.9	8.7	8.7	25.7	7.0	12.1	187.1
Magnesium Honeycomb		121.8	12.2	8.7	28.1	6.9	13.0	190.7
Graphite Composite Honeycomb		122.3	15.0	8.7	25.7	6.8	12.0	190.5
Aluminum Plate	10 inches	117.2	1.1	8.5	15.2	6.2	6.5	154.7
Magnesium Plate		116.2	1.9	8.5	16.7	6.2	7.1	156.6
Lockalloy Plate		116.3	49.1	10.2	17.9	6.3	7.7	207.5
Aluminum Honeycomb		124.9	7.8	8.5	18.0	6.7	9.0	174.9
Magnesium Honeycomb		121.8	12.0	8.5	19.4	6.6	9.6	177.9
Graphite Composite Honeycomb		122.3	17.2	8.5	18.0	6.6	9.1	181.7

TOTAL PROGRAM COSTS ($M) - FIVE YEARS - 151 FLIGHTS					
Configuration		Heat Shield Weight (lbs)	Payload Weight Penalty	Operational Costs	Total Program Costs
Subpanel	Fastener Spacing				
Direct Bond	N.A.	27,199	0	164.8	164.8
Aluminum Plate	5 inches	32,577	10.6	168.3	178.9
Magnesium Plate		31,602	3.6	173.8	177.4
Lockalloy Plate		30,092	0	205.2	205.2
Aluminum Honeycomb		32,158	7.3	187.1	194.4
Magnesium Honeycomb		30,882	0.5	190.7	191.2
Graphite Composite Honeycomb		30,994	0.6	190.5	191.1
Aluminum Plate	10 inches	37,803	76.9	154.7	231.6
Magnesium Plate		34,738	34.9	156.6	191.5
Lockalloy Plate		31,549	3.3	207.5	210.8
Aluminum Honeycomb		32,749	12.0	174.9	186.9
Magnesium Honeycomb		31,811	5.0	177.9	182.9
Graphite Composite Honeycomb		31,772	4.7	181.7	186.4

DISCUSSION OF PROGRAM RESULTS

Based on heat shield weight alone, the first six lowest weight ablator systems are ranked:

- Direct Bond . . . . . 27,199 lbs
- Lockalloy Subpanel at 5 inches . . . . . 30,092 lbs
- Magnesium Honeycomb at 5 inches . . . . . 30,882 lbs
- Graphite Composite Honeycomb at 5 inches . 30,994 lbs
- Lockalloy Subpanel at 10 inches . . . . . 31,549 lbs
- Magnesium Plate at 5 inches . . . . . 31,602 lbs

Based on total program costs, the first six ablator systems are ranked:

- Direct Bond $164.8M
- Magnesium Plate at 5 inches $177.4M
- Aluminum Plate at 5 inches $178.9M
- Magnesium Honeycomb at 10 inches $182.9M
- Graphite Composite Honeycomb at 10 inches . . $186.4M
- Aluminum Honeycomb at 10 inches $186.9M

The only repeaters in both lists are the direct bond and the magnesium plate subpanel at 5 inch attachment spacing.

While the direct bond ablator has the lowest heat shield weight and program cost, the concerns with dust control and minimization of turn-around time during the refurbishment period makes the second alternate attractive. Magnesium HM-21A subpanel at 5 inch fastener spacing is less than 5% heavier than the baselined RSI and only 8% higher in total program cost than the direct bond heat shield.

The decision of interim ablator usage should be made prior to Orbiter CDR to permit incorporation of ablator subpanel fastener hardware into the Orbiter structure. A late decision for interim ablator usage will make ablator heat shield attachment other than direct bond progressively more difficult and expensive.